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NASA CEV Reference GN&C Architecture



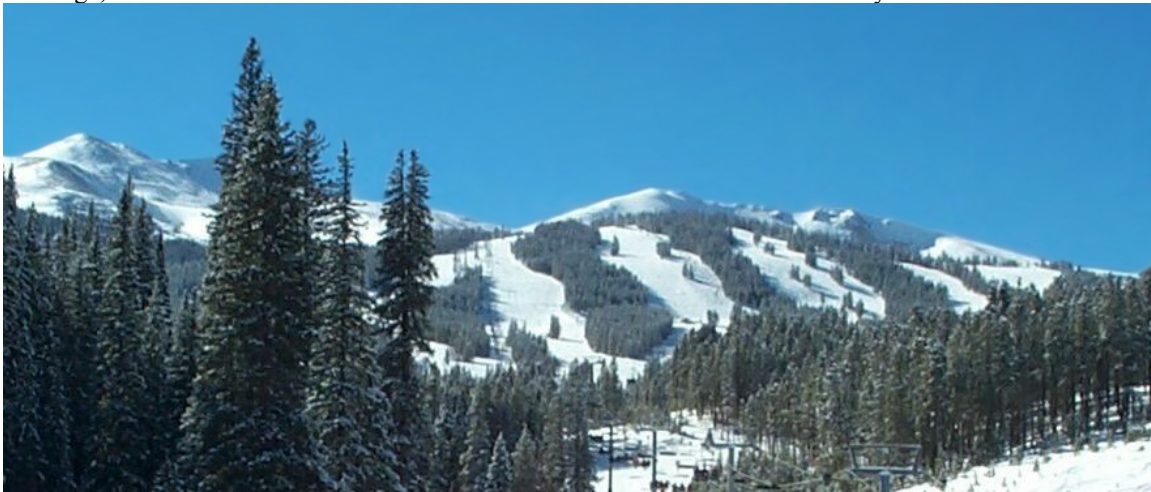
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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION (NASA) CREW EXPLORATION VEHICLE (CEV) REFERENCE GUIDANCE, NAVIGATION, AND CONTROL (GN&C) ARCHITECTURE

Scott Tamblyn, Heather Hinkel, Dave Saley

The Orion Crew Exploration Vehicle (CEV) will be the first human spacecraft built by NASA in almost 3 decades and will be the first vehicle to perform both Low Earth Orbit (LEO) missions and lunar missions since Apollo. The awesome challenge of designing a Guidance, Navigation, and Control (GN&C) system for this vehicle that satisfies all of its various mission requirements is countered by the opportunity to take advantage of the improvements in algorithms, software, sensors, and other related GN&C technology over this period. This paper describes the CEV GN&C reference architecture developed to support the overall NASA reference configuration and validate the driving requirements of the Constellation (Cx) Architecture Requirements Document (CARD, Reference 1) and the CEV System Requirements Document (SRD, Reference 2). The Orion GN&C team designed the reference architecture based on the functional allocation of GN&C roles and responsibilities of CEV with respect to the other Cx vehicles, such as the Crew Launch Vehicle (CLV), Earth Departure Stage (EDS), and Lunar Surface Area Module (LSAM), across all flight phases. The specific challenges and responsibilities of the CEV GN&C system from launch pad to touchdown will be introduced along with an overview of the navigation sensor suite, its redundancy management, and flight software (FSW) architecture. Sensors will be discussed in terms of range of operation, data utility within the navigation system, and rationale for selection. The software architecture is illustrated via block diagrams, commensurate with the design aspects.

INTRODUCTION

This paper summarizes NASA GN&C reference design architecture for the CEV, based on the Requirements Analysis Cycle (RAC) efforts of 2006, leading up to the Cx System Requirements Review (SRR). The purpose of this GN&C reference design was to assess the feasibility and correctness of the Cx and CEV requirements that drive CEV GN&C capabilities and to develop a sound engineering design baseline within NASA prior to prime contractor downselect and SRR.

The reference GN&C architecture described within this document addresses the driving requirements, the GN&C roles and responsibilities of the CEV, its navigation sensor suite and redundancy management scheme, and a candidate flight software architecture.

DRIVING REQUIREMENTS

Table 1 paraphrases the key requirements from the CEV SRD that drive the need, directly or indirectly, for targeting and GN&C capabilities onboard the CEV. Such capabilities are presumed to be implemented via onboard flight software integrated with the requisite sensors and effectors to provide the accuracy and precision required either to meet the functional requirement or the explicitly required performance specifications. The *italicized* requirements reflect the requirements owned and updated by the Flight Dynamics (FD) team and recommended to the Constellation Program Office for baselining prior to Cx SRR.

Table 1: DRIVING REQUIREMENTS ON CEV GN&C CAPABILITY

Requirement no.	Requirement
CV0038	The CEV return the crew to Earth during loss of communications with Earth during all mission phases.
<i>CV0039a</i>	<i>The CEV shall provide abort capability starting on the launch pad with the arming of the Launch Abort System through operations in LEO.</i>
<i>CV0039b</i>	<i>The CEV shall provide abort capability from Low Earth Orbit to arrival in the lunar reference orbit.</i>
CV0049	The CEV shall insert into LEO after being delivered to the mission-specific Earth Ascent Staging Target by the CLV.
CV0052	The CEV shall provide automated maneuvers to the CLV and to target abort landing locations.
CV0085	The CEV shall perform an automated deorbit maneuver from LEO.
CV0103	The CEV shall perform navigation for abort initiation and execution during loss of communications with Earth.
CV0106	The CEV shall perform the maneuver sequence to return from Low Lunar Orbit (LLO) to Earth.
<i>CV0107</i>	<i>The CEV shall calculate the maneuver targets for deorbit from LEO.</i>
<i>CV0108</i>	<i>The CEV shall calculate LLO navigation solutions for Trans-Earth Initiation (TEI) execution in less than 12 hours in the event of loss of communications with Earth.</i>
CV0109	The CEV shall perform trajectory correction maneuvers en route to Earth, returning from the Moon.
CV0110	The CEV shall independently provide navigation data of the CEV/CLV stack during ascent.
CV0111	The CEV shall independently provide navigation data of the integrated EDS/LSAM/CEV stack trajectory.
<i>CV0477</i>	<i>The CEV shall independently provide navigation data of the integrated LSAM/CEV stack trajectory.</i>
<i>CV0112</i>	<i>The active CEV shall perform relative navigation for rendezvous, proximity operations and docking with the target vehicle.</i>
<i>CV0121a</i>	<i>The CEV shall perform automatic execution of rendezvous, proximity operations, and docking under nominal conditions.</i>
<i>CV0124</i>	<i>The CEV shall perform automatic execution of separation, proximity and departure operations.</i>
CV0118	The CEV shall compute rendezvous maneuver targets while acting as the active vehicle during rendezvous.
CV0119	The CEV shall compute translational maneuver targets throughout all flight phases.
CV0131	The CEV shall perform rendezvous, approach proximity operations, and docking with the LSAM / EDS stack in LEO while acting as the active vehicle during rendezvous.
CV0132	The CEV shall perform contingency rendezvous and approach proximity operations with the LSAM in LLO with the unmanned CEV acting as the active vehicle.
<i>CV0471</i>	<i>The CEV shall provide for remote control and proximity and docking operations from the LSAM-AS for the Lunar Sortie and Lunar Outpost DRMs.</i>
<i>CV0133</i>	<i>The CEV shall support approach proximity operations and docking with the LSAM functioning as the active chaser vehicle in Low Lunar Orbit.</i>
CV0134	The CEV shall separate maneuver away from the unmanned LSAM Ascent Stage in LLO.
CV0450	The CEV shall separate and maneuver away from the unmanned LSAM in LLO prior to lunar descent.
<i>CV0135</i>	<i>The CEV shall provide attitude control of the mated CEV/LSAM Ascent Stage configuration upon completion of docking operations in LLO.</i>
CV0136	The CEV shall perform rendezvous, approach proximity operations, and docking with the ISS in LEO with the CEV acting as the active vehicle.
CV0137	The CEV shall separate and maneuver away from the ISS in LEO.

GN&C ROLES AND RESPONSIBILITIES

Figure 1 summarizes the events of interest for the lunar sortie Design Reference Mission (DRM) from launch through Earth entry in chronological order. The scenarios specific to the International Space Station (ISS) DRM in LEO are also shown, interspersed appropriately.

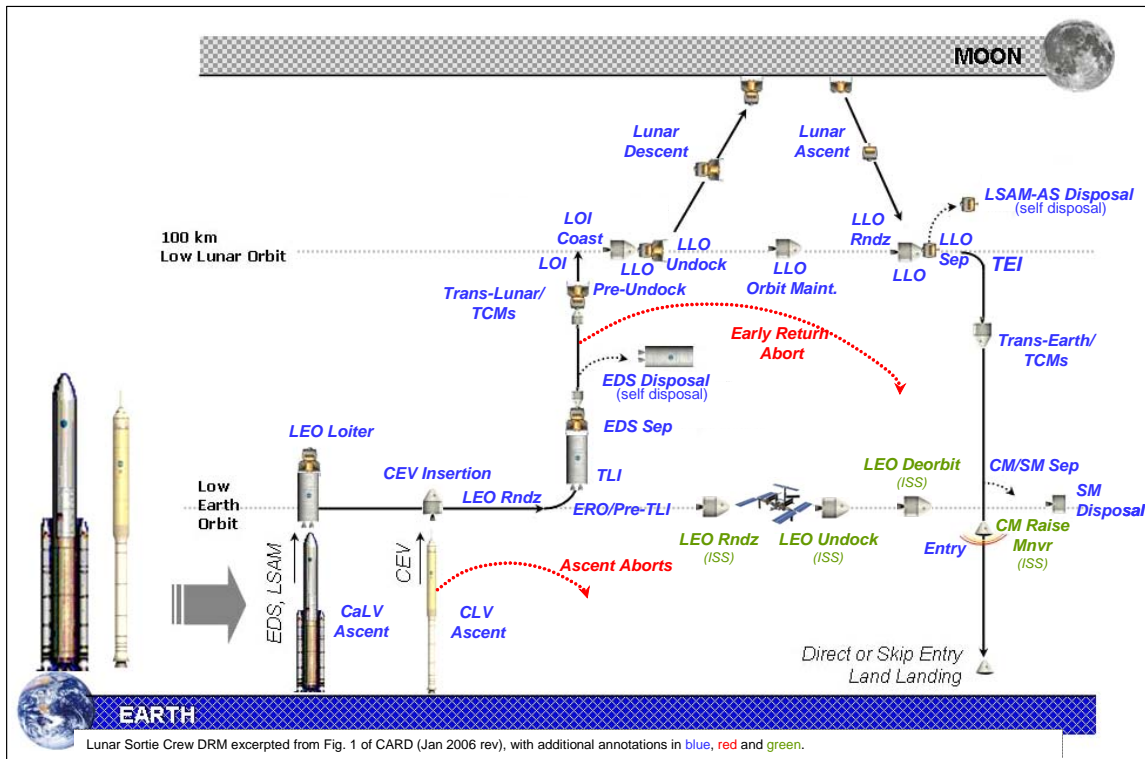


Figure 1: Events of Interest Overview

The following subsections provide an overview of the vehicles involved in the lunar sortie and ISS DRMs and a functional allocation of GN&C capabilities onboard the CEV based on the driving requirements and needs of the DRMs.

Crew Exploration Vehicle (CEV)

The CEV consists of the major elements shown in Figure 2.

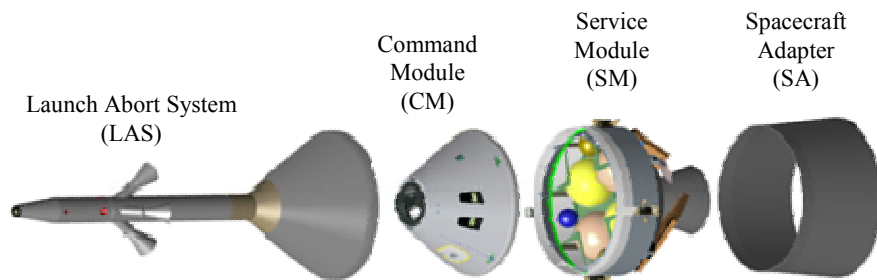


Figure 2: CEV Vehicle Components

From left to right those four elements are Launch Abort System (LAS), Crew Module (CM), Service Module (SM), and Spacecraft Adapter (SA). The LAS supports ascent aborts and is jettisoned prior to Earth orbit insertion. The SA provides the interface to the CLV and is also jettisoned from the rest of CEV before Earth orbit insertion. Most of the time the CEV will comprise the CM and SM. The crew resides in the CM. The SM provides CEV power, via deployable solar arrays, and translation and attitude control authority, via Reaction Control System (RCS) thrusters and a single main engine. The SM is jettisoned from the CM shortly before reentry.

Crew Launch Vehicle (CLV)

The CLV is used to launch the CEV. The CLV comprises two stages and the avionics required to support launch as shown in Figure 3. The CEV avionics will monitor the launch and support aborts.



Figure 3: CLV Vehicle Components

Lunar Surface Area Module (LSAM)

The LSAM is the vehicle that will take crew to and from the lunar surface, conceptualized in Figure 4 as consisting of an Ascent Stage (LSAM-AS) and a Descent Stage (LSAM-DS). The complete LSAM is mated with the CEV in Earth orbit and remains mated en route to the Moon and in LLO. The LSAM's RCS thrusters are responsible for mated LSAM/CEV stack control during translunar flight, including up to four scheduled Translational Correction Maneuvers (TCMs). The LSAM's main engines are responsible for the three braking and circularization maneuvers required to perform Lunar Orbit Insertion (LOI), which puts the stack in a stable LLO. The LSAM separates from the CEV and descends to the lunar surface through use of its LSAM-DS. The LSAM-AS separates from the LSAM-DS in order to return the crew into LLO. The LSAM-AS will nominally rendezvous and dock with the uncrewed CEV for crew and cargo transfer; upon separation from the CEV, the uncrewed LSAM-AS performs the necessary maneuvers for self-disposal to the lunar surface.

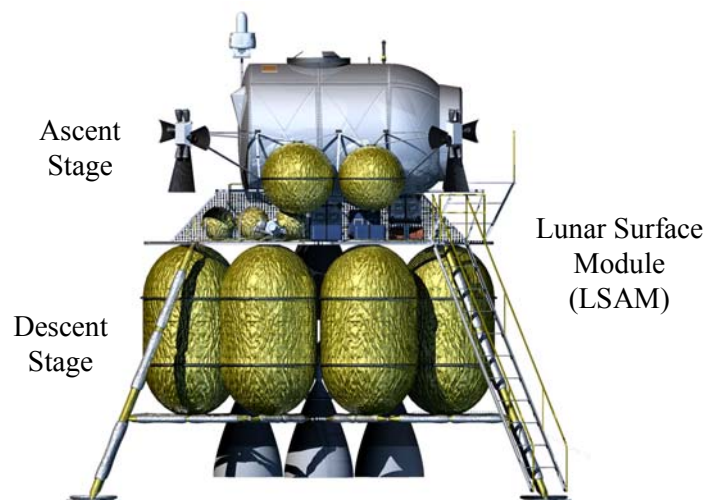


Figure 4: LSAM Vehicle Components

The LSAM, along with the EDS, is launched into Earth orbit atop the Cargo Launch Vehicle (CaLV), a launch vehicle different from the CEV's CLV.

Earth Departure Stage (EDS)

The EDS is actually the upper stage of the CaLV and is responsible for inserting itself and its unpowered LSAM payload into LEO. The CEV launches separately atop the CLV to rendezvous and dock with the LSAM/EDS stack in LEO. The primary function of the EDS is to perform the Translunar Injection (TLI) maneuver for the mated EDS/LSAM/CEV stack conceptualized in Figure 5. The EDS fires its main engines to put the mated stack on a lunar transfer trajectory. Soon after the TLI maneuver is complete, the CEV/LSAM stack separates from the EDS under LSAM control, while the EDS performs the necessary maneuvers for safely disposing of itself on the lunar surface. The EDS contains all the necessary avionics required for LEO insertion, attitude control, TLI execution, and self-disposal.

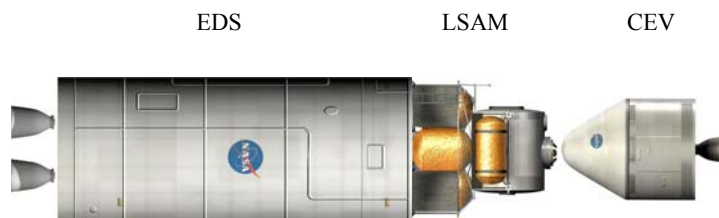


Figure 5: EDS/LSAM/CEV Stack Components

International Space Station (ISS)

Early missions of the CEV will support docking to the ISS as shown in Figure 6.

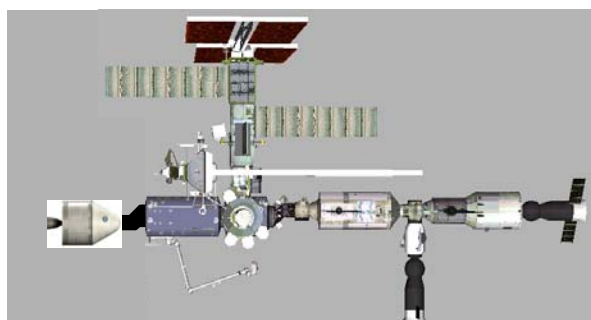
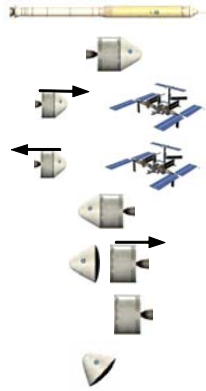


Figure 6: ISS/CEV Mated Configuration

GN&C Functional Allocation

This section shows which GN&C capabilities will be needed by the CEV in each of its key flight scenarios via a functional allocation table for each DRM. Figure 7 maps the targeting, navigation, guidance, and closed-loop flight control capabilities required of the CEV to support each flight event the ISS DRMs. The effector column denotes the scenarios and capacity in which the CEV's effectors will be used. Since most scenarios will have multiple vehicles present, this allocation technique shows when the CEV is responsible for executing the primary targeting and GN&C flight software and effecting it.

In each table a checkmark (✓) denotes a capability that will be present and executing nominally during an event, and an Ab (for abort) denotes a capability that, while present, will execute only during an abort or contingency situation occurring in that event. To show additional traceability, the marked table cells contain footnotes pointing to the requirements of Table 1 that drive those capabilities if they explicitly exist.



ISS	Tgt	Nav	Guid	Cntl	Eff
CLV Ascent	CV0119 CV0052	CV0110 CV0039a	Ab CV0039a	Ab CV0039a	Ab CV0039a
CEV Insertion	CV0119 CV0049	CV0049	CV0049	CV0049	CV0049
LEO Rndz	CV0118	CV0136	CV0136	CV0136	CV0136
LEO Undock		CV0137 CV0124	CV0137 CV0124	CV0137 CV0124	CV0137 CV0124
LEO Deorbit	CV0107	CV0085	CV0085	CV0085	CV0085
CM/SM Sep		✓	✓	✓	✓
SM Disposal					
CM Raise Mnv		✓	✓	✓	✓
Entry	Ab	✓	✓	✓	✓

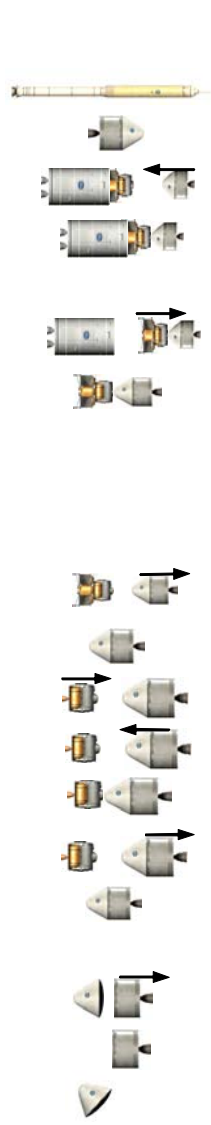
Figure 7: CEV GN&C Functional Allocation – ISS DRM

The marked cells are also color-coded in green and yellow. Green, which can be paired only with checkmarks, denotes a primary capability for flying the vehicle nominally. Yellow, which can be paired with either checkmarks or abort indicators, denotes a backup, contingency, or Situational Awareness (S.A.) capability. The yellow-checkmark combination means that the capability (e.g., navigation) is *executing* nominally but is serving only in a backup or S.A. capacity to the primary flight system. The yellow-abort combination means that the capability (e.g., guidance and control) *executes only in response* to an abort or contingency situation, potentially causing it to act as the driving flight system; e.g., an ascent abort causes a nominally passive CEV to depart from the CLV and become an actively flying vehicle on its own.

To the left of each functional allocation table is a graphical depiction of the vehicles involved in that scenario to aid in visualizing the operations. All graphics are artist concepts, as used in Figure 1, and are commonly seen throughout other Cx documents.

Figure 8 shows the analogous GN&C functional allocation for the lunar sortie DRM.

Reference 3 contains the full analysis description of these functional allocations.



Lunar Sortie	Tgt	Nav	Guid	Cntl	Eff
CLV Ascent	CV0119 CV0052	CV0110 CV0039a	Ab CV0039a	Ab CV0039a	Ab CV0039a
CEV Insertion	CV0119 CV0049	CV0049	CV0049	CV0049	CV0049
LEO Rndz	CV0118	CV0039b CV0112	CV0039b CV0131	CV0039b CV0131	CV0039b CV0131
ERO/Pre-TLI	Ab CV0039b	CV0111 CV0039b	Ab CV0039b	Ab CV0039b	Ab CV0039b
TLI	CV0119 CV0039b	CV0111 CV0039b	Ab CV0039b	Ab CV0039b	Ab CV0039b
EDS Sep	Ab CV0039b	CV0477 CV0039b	Ab CV0039b	Ab CV0039b	Ab CV0039b
Translunar/TCMs	CV0119 CV0039b	CV0477 CV0039b	Ab CV0039b	Ab CV0039b	Ab CV0039b
LOI (-1,-2,-3)	CV0119 CV0039b	CV0477 CV0039b	Ab CV0039b	Ab CV0039b	Ab CV0039b
LOI Coast	CV0119 CV0039b	CV0477 CV0039b	Ab CV0039b	Ab CV0039b	Ab CV0039b
LLO Pre-Undock	CV0119 CV008	CV0477 CV008	Ab CV008	Ab CV008	Ab CV008
LLO Undock		CV0450 CV0124	CV0450 CV0124	CV0450 CV0124	CV0450 CV0124
LLO Orbit Maint.	CV0119	CV0108 CV0106	CV0106	CV0106	CV0106
LLO Rndz (Nom)		CV0133	CV0133	CV0133	CV0133
LLO Rndz (Cont)	CV0118	CV0132 CV0112	CV0132 CV0121a	CV0132 CV0121a	CV0132 CV0121a
LLO		CV0135	CV0135	CV0135	CV0135
LLO Sep		CV0124 CV0134	CV0124 CV0134	CV0124 CV0134	CV0124 CV0134
TEI	CV0119	CV0108 CV0106	CV0106	CV0106	CV0106
TransEarth/TCMs	CV0119	CV0109	CV0109	CV0109	CV0109
CM/SM Sep		√	√	√	√
SM Disposal					
Entry	Ab	√	√	√	√

Figure 8: CEV GN&C Functional Allocation – Lunar Sortie

NAVIGATION SENSORS

The CEV uses a suite of inertial and relative sensors for determining its navigation state. Inertial sensors aid the GN&C system in the determination of the absolute state of the vehicle with respect to an inertial frame, typically fixed in either the Earth or the Moon. Relative sensors aid the GN&C system in the determination of the relative state of the vehicle with respect to a target vehicle-fixed frame [such as a rotating, Local Vertical/Local Horizontal (LVLH) frame during rendezvous and proximity operations] or a line-of-sight reference (such as the ground in the case of landing sensors).

Figure 9 provides an overview of the CEV GN&C sensors described in this section with reference vendors, models, and numbers of units also detailed. The blue background groups the inertial sensors, the yellow background groups the relative sensors, and the green shows the overlap between the two; i.e., sensors that serve both purposes.

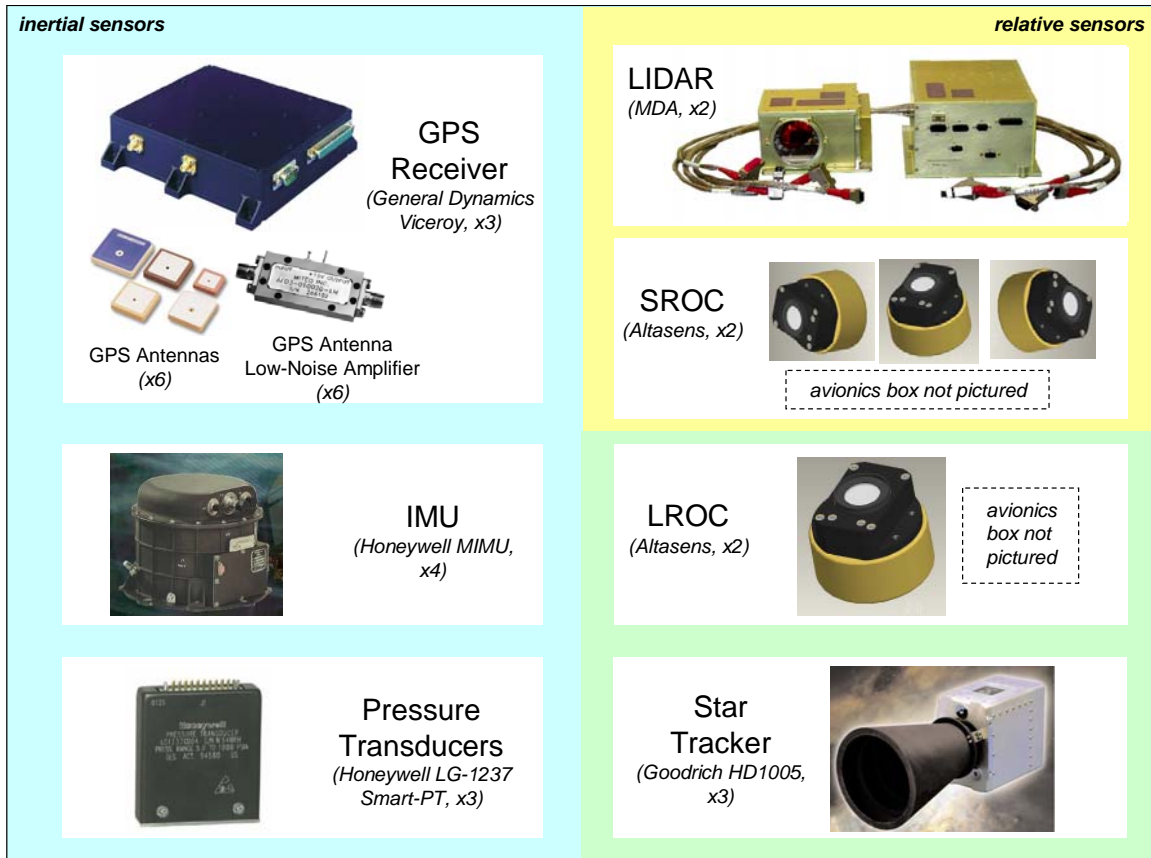


Figure 9: CEV GN&C Sensors

Figure 10 shows the ranges of applicability for each of the relative sensors during Rendezvous, Proximity Operations, and Docking (RPOD). The chart also shows the ranges at which sensor data yields range data only; bearing data only; range and bearing data necessary for three Degree-of-Freedom (3-DOF) state determination; and range, bearing, and relative attitude data necessary for 6-DOF state determination.

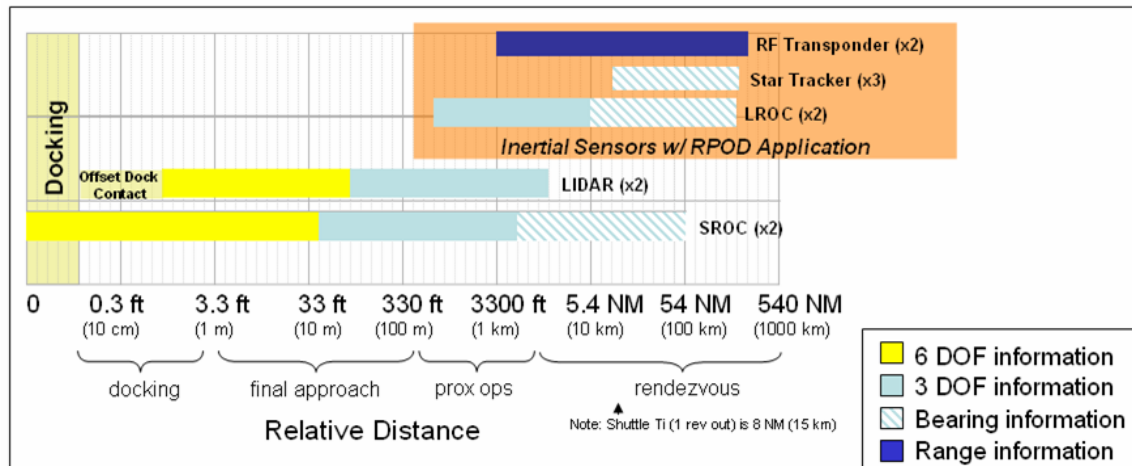


Figure 10: Sensor Ranges of Applicability During RPOD

This sensor suite is the reference design submitted for the CEV project's CEV Reference Configuration 3 (CRC3). The Redundancy Management (RM) scheme planned is designed to meet the 2-Fault Tolerance (2FT) requirement in all flight phases and provide the greatest flexibility for navigation use. In addition to the sensors shown here, a radio frequency (RF) transponder book kept by the avionics system will be used for navigation when Global Positioning System (GPS) is not available.

Global Positioning System (GPS)

A GPS sensor system includes both the receiver and the antennas to provide onboard inertial position and velocity state vector updates during Earth ascent, LEO, and Earth entry phases of flight. Two antennas are required per receiver for near-full coverage of the GPS constellation of satellites (typically an upper and a lower antenna for seeing the upper and lower hemispheres of the sky). To alleviate signal losses in the cable runs between the antennas and the receivers, each antenna also requires a low-power 38-dB low-noise amplifier.

The CRC3 reference design contains three General Dynamics Viceroy model GPS receivers with two antennas each (six total).

Inertial Measurement Unit (IMU)

The IMU contains 3-axis strap-down-rate gyros to measure vehicle body attitude rates with respect to the inertial frame and 3-axis accelerometers to measure vehicle body accelerations with respect to the inertial frame. This inertial data is required for onboard navigation to dead reckon (propagate) between (or in the absence of) position, velocity, and attitude state updates from the ground segment.

The CRC3 reference design contains four Honeywell MIMU model IMUs.

Star Tracker (ST)

Through sighting on stars, the ST provides the highly accurate inertial attitude measurements required to update the IMU periodically throughout the mission via an IMU Align (the frequency of this update is a function of the IMU's drift/performance characteristics). The primary output of the ST is an inertial-to-body attitude quaternion.

In addition to inertial sensing, the ST also doubles as a rendezvous sensor via a Target Track feature that allows it to measure the relative bearing (azimuth and elevation) of the target vehicle (the EDS/LSAM stack, ISS, or LSAM-AS) at ranges far outside the field of view of the Automated Rendezvous and Docking (AR&D) sensors. During the far-field rendezvous [from 400 nautical miles (nmi) to 20 nmi, approximately], ST-measured bearing and transponder-measured range are the key measurements for determining the relative state of the vehicle.

The CRC3 reference design contains three Goodrich HD1005 model STs. This ST is one of the few that provides the required Target Track feature. Due to the desired mounting location of the STs on the CM, they must be thermally protected with Mylar blankets.

Pressure Transducers

Pressure transducers robustly provide highly accurate atmospheric pressure data (pressure/mean sea level altitude) that is critical for guidance and control during Earth entry, including chute deployment and landing. Accuracy and data availability in the altitude channel is superior to that measured and maintained by the GPS and IMU, hence the design decision to include this sensor in the suite.

The CRC3 reference design contains three Honeywell LG-1237 Smart PT model pressure transducers.

Light Detection and Ranging (LIDAR)

The LIDAR sensor contains a laser that provides time-of-flight-based relative range, bearing, and attitude measurements during RPOD flight phases via direct signal returns from target-mounted retroreflectors. The LIDAR

measurements support manual piloting, remote control, and AR&D operations. LIDAR functionality is independent of orbital lighting once the target is initially acquired.

The CRC3 reference design contains two MDA LIDARs that have flight test heritage onboard the XSS-11 satellite.

Long-range Optical Camera (LROC)

The LROC capabilities serve two GN&C functions. Primarily, the LROC provides imaging required to perform Celestial Navigation (CelNav) during both cislunar transit and LLO to give the vehicle an onboard navigation capability while outside the influence of GPS. While cislunar, the LROC provides an image of a star field observation, and the CelNav algorithm uses the ST-provided inertial attitude to reduce the star image back into the inertial frame to compute a state. While in LLO, the LROC can track the lunar surface and, via surface feature recognition and CelNav algorithms, determine an inertial state based on bearing measurements. The surface feature tracking algorithm may also be able to use apparent diameter (from the LROC image) as a range measurement to further aid state determination.

During rendezvous operations, the LROC provides relative range and bearing measurements to augment similar measurements taken by the other AR&D sensors.

The CRC3 reference design contains two Altasens LROCs. Each LROC consists of a single 2.0-megapixel optical sensor unit and a separate avionics box containing a video processing unit, a communications interface, and timing and control electronics.

Short-range Optical Camera (SROC)

Processed images of the target vehicle/docking target array from the SROC provide relative range, bearing, and attitude measurement data during RPOD operations to support manual piloting, remote control, and AR&D operations. In addition to being a data sensor, the SROC provides motion imagery for the onboard crew, the remote vehicle crew, and the Mission Planning, Training, and Flight Operations (MPTFO) elements. SROC use is limited to orbital daylight or while within range of the illuminated target array on target vehicle. The illumination of the target array is achieved via a separate illuminator/docking light comprising light emitting diodes (LEDs) that will provide target illumination out to approximately 650 feet (200 meters).

The CRC3 reference design contains two Altasens SROCs. Each SROC unit consists of three 2.0-megapixel optical sensor units and a separate avionics box to handle the multicamera multiplexing, video processing, timing, and control electronics and communications required for the optical triad. The three optical sensors are like those of the LROC, but they are bundled together in a single assembly. The orientation of each of the three cameras with respect to the assembly can be configured premission to the optimal geometry required for RPOD operations. While the triad architecture of cameras for a single SROC provides additional levels of capability and redundancy, the SROC by itself is still a zero-fault-tolerant sensor by virtue of its single avionics box; therefore, two SROCs are still needed in the reference design for system redundancy.

Radio Frequency (RF) Transponder

These transponders are actually the same ones used by the communications (comm) system and, therefore, bookkept by the comm system. Their dual role in the GN&C arena, however, is significant enough to warrant discussion here. The transponders are the conduit for receiving ground-tracked, inertial position, and velocity (not attitude) state uplinks from MPTFO elements. In addition, the transponders provide Doppler relative-range and range-rate measurement during rendezvous operations (out to 400 nmi) through pinging a sister transponder on the target vehicle.

Landing Sensors

The CM will use a range-above-ground relative sensor to provide accurate and precise altitude information just prior to landing. This information will be used to trigger a capsule landing orientation (roll) maneuver aligning the crew with the horizontal velocity at touchdown, if desired, and to trigger the firing of CM-mounted retrorocket jets

designed to minimize vertical velocity and impact loads at touchdown. The baselined sensor for this function is the MSP '98 Landing Radar by Honeywell, a 4-beam radar altimeter originally adapted for use on a Mars lander that can provide altitude and limited horizontal velocity information for an Earth landing without ambient lighting. Two redundant radar avionics units will be integrated into the CM; both are connected to a single antenna cluster of four integrated antennas oriented down. Antennas have a very high reliability relative to the radar avionics. Like the retrolanding jets, use of this radar will require release of the CM heat shield because it is unlikely that it will be able to work effectively, sending and receiving pulses, through the material selected for the heat shield. An estimate of horizontal velocity as made available by this sensor will also be used to determine if the CM is landing on a modest slope and potentially to adjust the crewmembers' impact orientation appropriately.

The GN&C logic and systems will communicate with the radar altimeter units via a software Subsystem Operating Program (SOP) as described in the flight software subsection and will monitor the system health of each unit, send configuration commands to each, and receive range (altitude) and estimated horizontal velocity from the active radar system.

Backup Sensors

As part of the emergency entry system design, the CM may also use backup pressure altitude sensors, independent of the flight computer and enabled by manual command, for triggering critical landing system events, specifically nose cone release, mortar firings for drogue and main chute deployment, and heat shield release. Otherwise an emergency entry system backup for triggering chute deployment could simply involve manual switch throws.

Other possible backup sensors to support an emergency entry system landing could include sending one of the IMU's raw outputs to a separate attitude display that a crew (or flight computer that has lost its absolute navigation orientation) could use to null out attitude rates if all primary attitude navigation is lost. An automated system that has lost its navigation state could also use an emergency backup 3-axis accelerometer package to sense deceleration just after entry interface and orient the capsule appropriately (heat shield first) prior to commanding a ballistic spin-up.

An even simpler approach for a possible emergency entry system could involve placing some attitude reference marks on the windows that the crew could use to orient the capsule, via manual RCS commands, prior to entry if all flight computers fail. This approach uses very simple techniques available to both Apollo and Soyuz.

Other capabilities are under consideration as part of an Entry Emergency Mode capability, including (1) a simple backup flight computer with a significantly reduced instruction set (just sufficient to orient the capsule, spin it up, and deploy chutes) or (2) similar instructions existing in a safe mode software configuration for a rebooted primary computer after a total computer failure.

Sensor Locations

Figures 11, 12, and 13 show some conceptual layouts of where key inertial and relative navigation sensors could be installed on the CEV in order to achieve the required fields of view and lines of sight during the DRMs.

One of the key placement changes made during the CRC3 design cycle was the reorientation of ST 3 so that it points perpendicular to the capsule's longitudinal (X-body) axis rather than perpendicular to the capsule sidewall as STs 1 and 2 do. This reorientation will allow ST 3 to see stars while the CEV is in Orbit Maintenance mode in LLO, holding an LVLH attitude pointing radially down at the lunar surface; this alleviates concerns that the fields of view of STs 1 and 2 will be obscured by the Moon.

Although not explicitly shown in the figures, another key placement constraint the GN&C team imposed on the layout was the placement of all IMUs and STs on a common structure to serve as a navigation base to minimize alignment errors between those sensors due to vehicle flex caused by thermal and vibration effects.

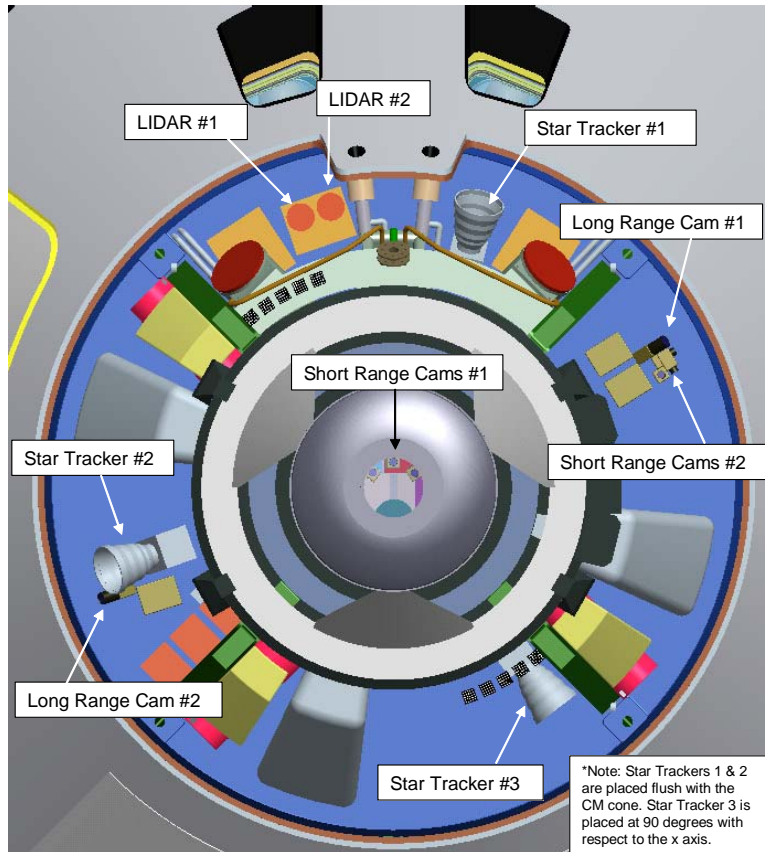


Figure 11: Sensor Locations (Centerline View)

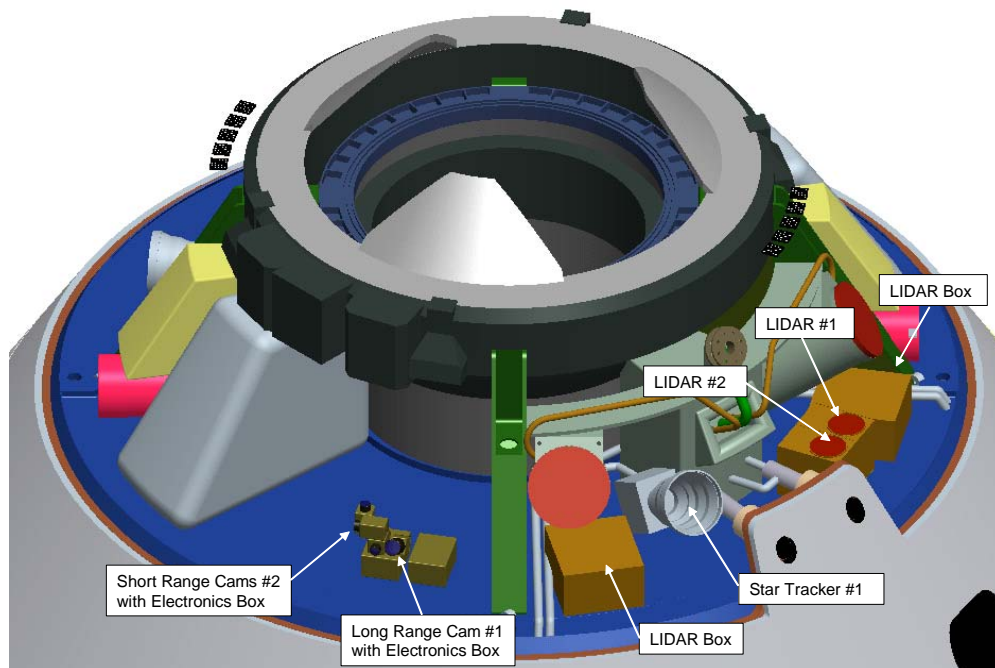


Figure 12: Sensor Locations (View 1)

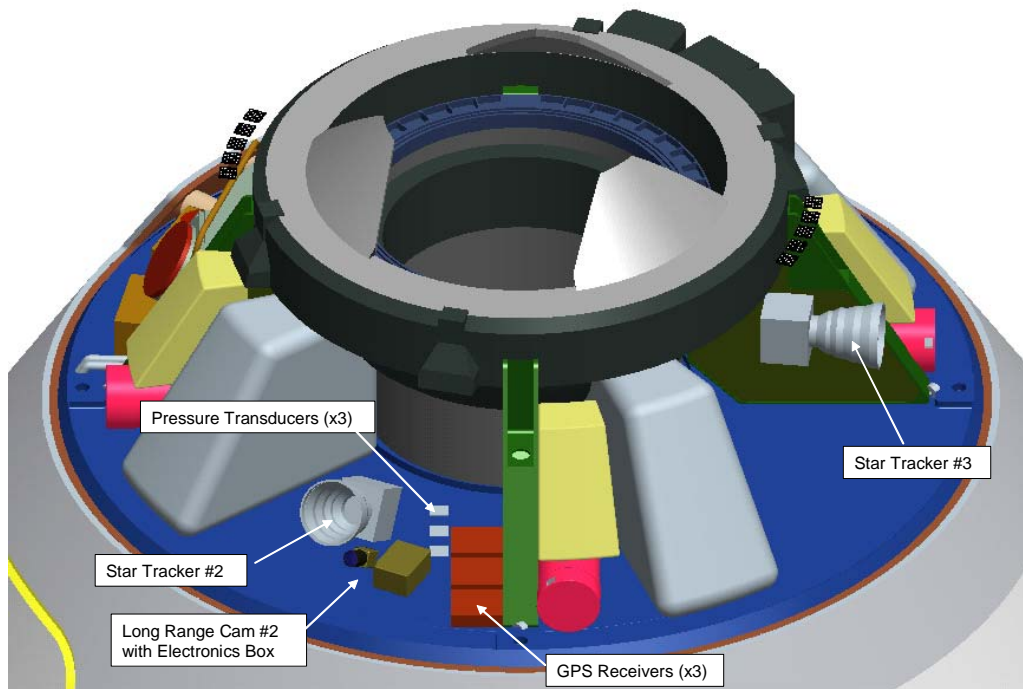


Figure 13: Sensor Locations (View 2)

REDUNDANCY MANAGEMENT

Figure 14 shows the block diagram for a candidate navigation architecture. This scheme provides a mechanism to meet the RM and FT objectives of all stages: ascent, entry, LEO, lunar transit, and LLO. Four strings of navigation states are maintained using four identical filters, one of which is configured as an IMU-only propagator. All sensors can be cross-strapped to all filters. Each filter configuration and the desirability of its filtered state output for use are configurable via initialization load (I-load) and flight mode settings. These I-loads would determine, per flight phase, how many filtered state are maintained, which sensors are used, and what sensor outputs are sent to which filters. Fault Detection, Isolation, and Recovery (FDIR) and sensor selection occur before filter input, and one set of sensors is preselected for insertion into a prime designated filter. The prime filter gets the best measurements available based on which selection scheme is used; e.g., weighted average, midvalue select. Two other strings maintain the remaining unique sensors; a quality rating relative to the prime string can be assessed, if desired. One IMU is used as a clean propagator string based on the best performing IMU data with no other data inputs other than periodic prime state updates. Flight phase dictates the number of filter strings to be considered (1, 2, 3, or 4) for prime selection; however, all are maintained. The strings can be monitored by the ground. Strings that are not used for the navigation state during some phases can be used for real-time testing of other redundancy modes for future upgrades. All strings can be reinitialized with the navigation state at any time (similar to ISS's scheme). If a sensor fails, the prime string will automatically select another one; the bad sensor will be removed, and a default sensor will be cross-strapped to the third string. Logic for the subsequent recovery and reintegration of a failed sensor will reside in the FDIR. In this case, entry can have the choice of eliminating the third string and incorporating the fourth (IMU-only propagator) for voting or keeping a cross-strapped string in the voting loop and using the fourth string as a quality check on the three voted strings. The prime string could be designed to be either *sensor dependent* (i.e., all data from the selected sensor goes to that string) or *variable independent* (i.e., best values of each variable go to the prime regardless of sensor).

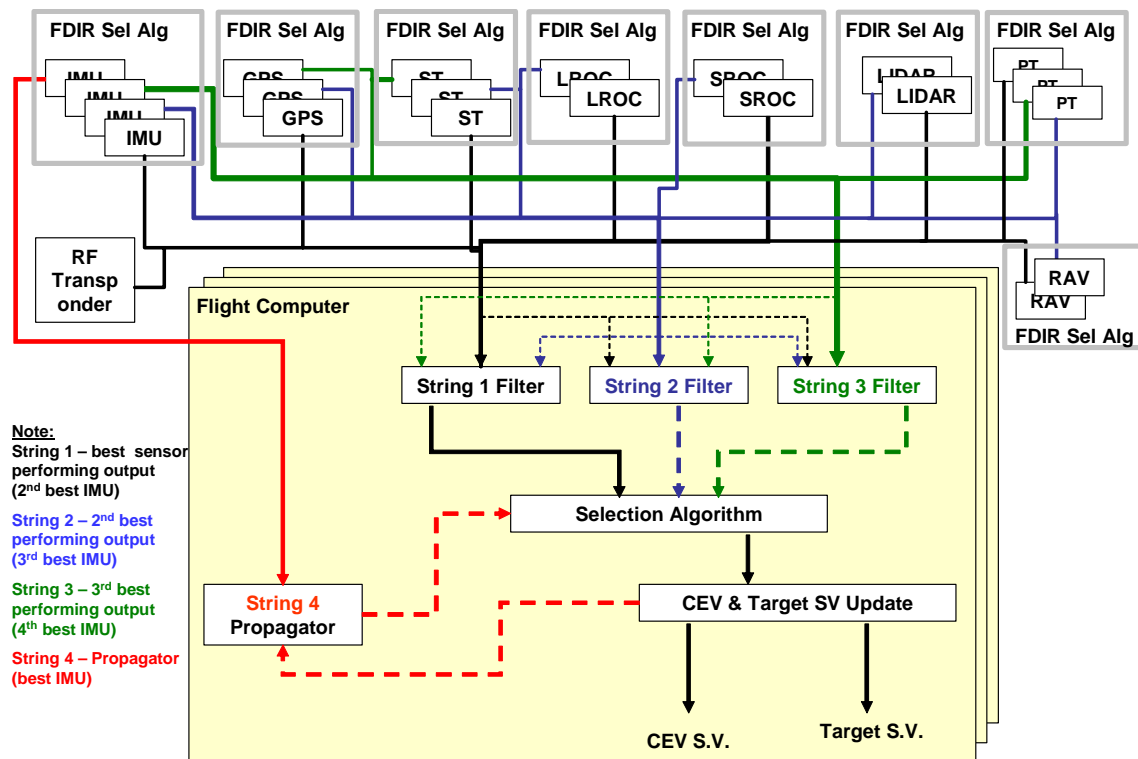


Figure 14: CEV Navigation Sensor Redundancy Management Architecture

During ascent all four strings are initialized from clean IMU measurements, and their solutions would be propagated solutions without other sensor measurements. The navigation state would be output based on the prime string, which could be set to take best IMU measurements of all four IMUs or just a specific selected one. Prior to the abort modes, the sensor data would be allowed into the filters after passing through FDIR and sensor selection.

During orbit, prox-ops, docking, LEO, and LLO, each sensor could be turned on or off or mapped to a different string with the best measurements going to the prime; the other strings are maintained as a quality check on the system. Since the prime is always used as the navigation state during these phases, the concern that switching between filters causes discontinuities (i.e., data jumps) is not an issue. Each string could be monitored by the ground, and as faults are detected the degraded or failed sensors could be removed from the strings and replaced, via software reconfiguration commands, by other valid sensors. Reconfiguration could be done either automatically or manually by the ground or crew, as required. For LROC, SROC, and LIDAR, each unit could be reconfigured to a different filter string, with the prime selecting the best values based on the preselection criteria. If only one of each sensor type is powered, the other filter strings could be used as straight propagators; alternatively those sensors could also be cross-strapped to the other filters for that flight phase. The selection filter would also allow for a selected navigation state to be provided back to the filter strings for reinitialization and resynchronization. The selection criteria possibilities are limitless and could be tested in real time if all four filters are always running.

For entry, having all necessary redundant sensors powered on with four navigation filters and a selection algorithm for determining the best state solution is an excellent redundant system. The fourth IMU-only/propagator string would be updated just prior to entry and would propagate to the ground. One string could completely diverge, and the performance of the other three strings would still meet the landing accuracy objectives. FDIR and sensor selection should pick up any known anomalies so that the prime filter can provide the best solution all the way to ground; the other three strings provide verification that the prime has not diverged. Should the prime diverge, the selection filter would select another filter string. In the event a second string fails, the IMU-propagated solution would complete the task, albeit with an accuracy that may impact the landing accuracy objectives.

The primary advantage of this design is that it gives the most flexibility; sensor/filter reconfiguration can be made on the fly if needed. Although this flexibility allows numerous sensor/filter configuration permutations, only the ones planned for nominal and contingency flight operations would require certification. If additional strings and/or functionality is added in later CEV system upgrades, those performance permutations can be verified using flight data from previous missions. The I-loadable, cross-strapped nature of the design would, therefore, foster the ability to improve navigation accuracy in future missions as more flight experience and flight data are accumulated. Such flexibility and ease of future adaptation, however, come at the expense of a complex implementation.

Reference 4 contains the full analysis description of this and other candidate RM schemes and the trades involved.

GN&C FLIGHT SOFTWARE ARCHITECTURE

Figure 15 shows a functional block diagram overview of the CEV GN&C system and its interfaces to its own sensors, effectors, and external Cx elements. The central block in the diagram functionally represents the Flight Critical Processor (FCP) on which the avionics and GN&C FSW resides and executes.

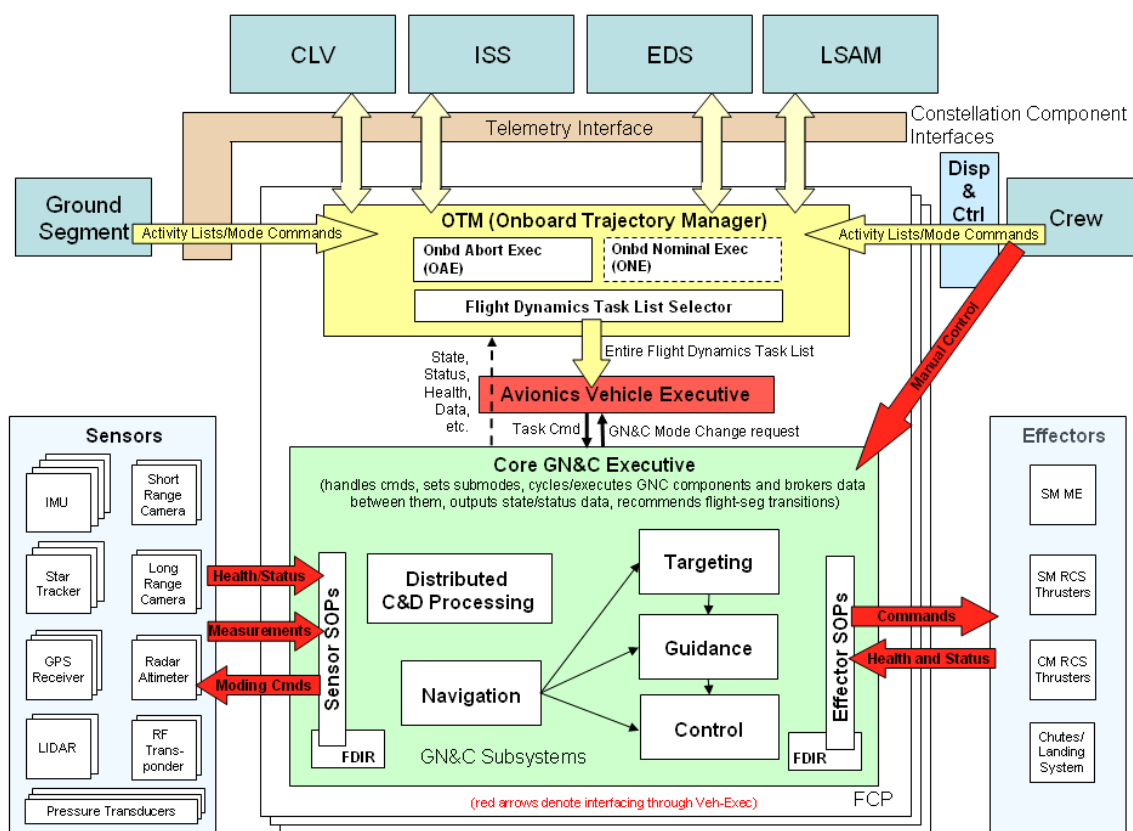


Figure 15: CEV GN&C Software Architecture

This section describes the software elements internal to the CEV FCP that are relevant to FD and GN&C. The classical system elements of top-level avionics Vehicle Executive FSW and FD core GN&C FSW are both represented. In response to the CARD and CEV SRD requirements for automated and autonomous FD events and responses, a new class of software subsystem, the Onboard Trajectory Manager (OTM), has been envisioned. The OTM is the FD slice of a vehicle-wide, systems-wide avionics manager known as the Vehicle Systems Manager (VSM). Although the VSM reference architecture is concurrently being designed by the CEV avionics group, the FD discipline understands VSM to be the top-level avionics application that schedules and executes all FCP

applications, including its own avionics Vehicle Executive and the two major FD components, the OTM and core GN&C. The VSM has integrated oversight responsibility of all of the vehicle's subsystems. This section explores some of the key details of the OTM, the Vehicle Executive, the GN&C Executive, and the GN&C sensor and effector SOPs. Additional details regarding the core GN&C blocks themselves, flight-phase dependent algorithms, and key engineering parameters flowing between them can be found in Reference 5.

Onboard Trajectory Manager (OTM)

The OTM is essentially a trajectory planning application that contains three main subsystem components: the Onboard Abort Executive (OAE), the Onboard Nominal Executive (ONE), and the FD Task List Selector (FDTLS). The OTM can receive such external inputs as activity lists and mode commands from the ground and crew systems, internal inputs from its own subsystems (OAE, ONE, and FDTLS), and navigation state and status feedback from core GN&C. The OTM's primary output is an FD Task List, which is simply a queue of FD tasks intended to be executed in sequence by core GN&C. The task data may include such various trajectory parameters as vehicle position, velocity, attitude, and attitude rates. The OTM and core GN&C exchange information via the Vehicle Executive.

The OAE is intended to plan and manage such automated abort trajectories as ascent aborts. The ONE is intended to plan and manage automated nominal trajectories; e.g., during AR&D operations. Both the OAE and ONE output FD Task Lists to the OTM's FDTLS based on their planning algorithms and operational flight rules and constraints. The FDTLS is then responsible for prioritizing all provided FD Task Lists, allowing for override considerations and factoring in cross-cutting systems issues from VSM, to output a single FD Task List to the Vehicle Executive. Cross-cutting systems issues may include resolving power resource contentions, blowing pyrotechnic devices, activating buses, and stirring propellant tanks. A specific example is the proverbial duck-in-the-windshield scenario in which the vehicle hits a duck during ascent, thus causing a life-critical cabin leak worthy of mission abort. Being a life-support systems problem, the FD subsystems of OAE, ONE, and core GN&C would be oblivious to the problem and press onward with the mission as if nothing were wrong. VSM, however, would be aware of the problem and could command an ascent abort to the OTM's FDTLS, which would take priority and result in a new abort-related FD Task List being sent to the Vehicle Executive for execution by core GN&C.

As unexpected FD scenarios or contingencies arise that would impact the FD Task List, which would most likely be reworked and/or reordered by the OAE, the OTM can simply pass the updates to the Vehicle Executive for execution.

A prototype OAE application has been developed to support ascent abort trade studies under FD Task Description Sheet (TDS) 04-012, Integrated Ascent and Abort Characterization and Sensitivity Study (Reference 6). More details on its internal architecture and functionality can be found there.

Vehicle Executive

The Vehicle Executive subsystem is envisioned as the avionics VSM engine responsible for executing all of the vehicle's subsystem executives; e.g., GN&C, power, environmental control, life support. FD data that is required by OTM and GN&C is brokered through the Vehicle Executive. The Vehicle Executive receives GN&C state and health data from the GN&C Executive to relay to the OTM; it also receives prioritized FD Tasks Lists from the OTM (courtesy of its FDTLS) with which to command the GN&C Executive. Rather than simply passing the entire FD Task List to the GN&C Executive, the Vehicle Executive doles out the task commands contained in the FD Task List, task by task in real time. Task commands may be simple mode commands or more complex commands with argument data; e.g., a commanded attitude or velocity. This design allows the GN&C Executive to deal with only one command at a time rather than having to take in the entire FD Task List at once and dole it out internally. This approach leverages a queuing capability envisioned already to be part of the Vehicle Executive, thus removing the requirement from GN&C and simplifying the integrated design.

The Vehicle Executive is also responsible for gathering all the input data required by core GN&C (e.g., sensor data messages, system state data, commands from the FD Task List) and providing it in a time-homogenous fashion. Likewise, it is also responsible for gathering output data from core GN&C (e.g., effector command data, vehicle navigation-state data) and relaying it to the proper hardware and software users in a time-homogenous fashion.

The Vehicle Executive contains vehicle mode transition rules for determining when to advance from one task to the next in the FD Task List based on key GN&C task-completion data from the GN&C Executive; e.g., timers, key altitudes and navigation states, events, submode completion flags. As such, the Vehicle Executive is responsible for coordinating and executing the flight phase and flight segment transitions across the vehicle's subsystem. The driver for handling phase, segment, and mode transitions in the Vehicle Executive rather than the GN&C Executive is that these transitions are of interest and consequence to the entire vehicle and not just GN&C.

GN&C Executive

The GN&C Executive is the interface for all Inputs and Outputs (I/O) between core GN&C (which includes sensor and effector SOPs) and the external world (which includes the sensor effector hardware).

Figure 15 shows sensor and effector I/O in a functional sense via the red arrows to denote that the data will actually pass between the GN&C Executive and the hardware via the Vehicle Executive. The Vehicle Executive serves as the mechanism for getting the data into the proper avionics common data areas (CDAs) for driving the hardware since avionics will be responsible for all I/O system services; e.g., 1553 data buses, RS-422, 1394a.

When the GN&C Executive receives the external world's inputs, it is responsible for checking and enforcing the validity of commanded mode transitions to ensure robustness, correctness, and consistency within GN&C to protect against instances of invalid FD Task List commands. As discussed in the Vehicle Executive section, the GN&C Executive handles only one task command at a time to simplify interfacing and testing. The GN&C Executive is responsible for setting the submodes of its individual components and coordinating and brokering data between them, which includes sensor SOPs, targeting, navigation, guidance, control, and effector SOPs. An example of such data brokering is mapping navigation outputs to guidance inputs. Part of the GN&C Executive's data brokering responsibilities is ensuring the time homogeneity of the overall GN&C outputs to the Vehicle Executive just as the Vehicle Executive ensured the inputs.

Figure 16 shows a functional flow of key data between the GN&C subsystems under the purview of the GN&C Executive.

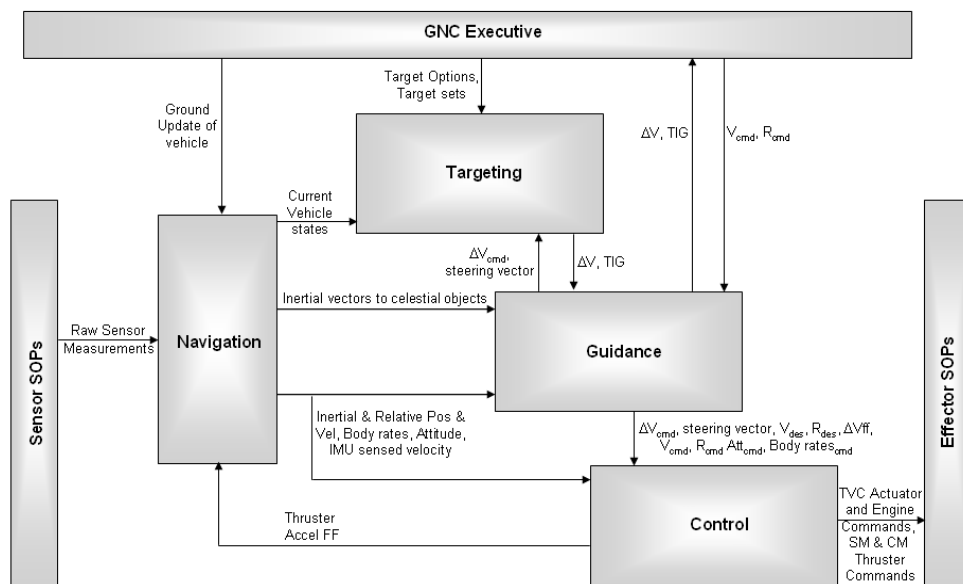


Figure 16: CEV Core GN&C Internal Interactions

The GN&C Executive executes its GN&C subsystems based on flight phase, segment, and mode data received from the FD Task List command and the corresponding submodes of each GN&C subsystem. Subsystem execution may be a function of cyclic rate group; e.g., guidance subsystems may run at a lower frequency while flight control

subsystems run at a higher frequency. Rate group requirements, while immature at this point in the design process, may require multiple GN&C Executives running as separate tasks in the avionics data management system.

GN&C Subsystem Operating Programs (SOPs)

Each GN&C sensor and effector type has its own software subsystem dedicated to receiving raw data from the hardware; processing it to extract and generate the engineering measurement, health, and status information to pass on to the core GN&C algorithms; and relaying initialization and moding commands to the hardware from GN&C. Such a software subsystem is termed a SOP, which is the heritage term from the shuttle's software architecture implementation.

Each SOP receives the data from all redundant hardware units of a similar type (e.g., a GPS SOP receives the GPS raw data from each of the three GPS units) and performs whatever FDIR functions may be required to assess the health of the hardware and react to anomalies where action can be taken to ensure hardware and subsystem integrity and safety. The FDIR may include such functions as reasonableness checks to ensure data integrity and selection algorithms to quantify and qualify the performance of each redundant hardware unit; e.g., to determine which unit is performing best and should be designated prime. The selection algorithm functionality is key to a SOP's ability to support RM. Reference 4 describes the recommended reference design for how a sensor SOP's fault tolerance, FDIR, and RM capabilities interact with core GN&C's navigation filters and navigation state selection algorithms.

GN&C SOPs are considered part of the core GN&C system but use avionics (CDA) and hardware interfacing protocols (e.g., MIL-STD-1553, RS-232) for brokering the data between the GN&C FSW and the hardware. In other words, the SOP would not directly communicate with the hardware interface for data receipt and commanding; the SOP would communicate through the GN&C Executive's software interface to the avionics CDA mechanisms by way of the Vehicle Executive and/or VSM in order for avionics to do the direct communication with hardware; e.g., in the case of a 1553 hardware interface, GN&C would not do any direct 1553 bus commanding but would relay the appropriately packed messages to avionics resources for avionics to schedule and control the 1553 bus commanding.

CONCLUSION

The reference GN&C architecture summarized in this document addresses the driving requirements, the GN&C roles and responsibilities of the CEV, its navigation sensor suite and redundancy management scheme, and a candidate flight software architecture. This reference design was done independently by NASA during the CEV prime contractor downselect period, Phase 1, in an effort to make NASA smarter customers after downselect and good stewards of the CEV requirements. The effort behind this reference design will help NASA understand the design trades and issues of the requirements when merging with the prime contractor and overseeing development of the CEV.

ACKNOWLEDGMENT

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